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Lunar Human Landing System Architecture Tradespace Modeling

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Abstract

A renewed interest in lunar exploration with the focus on establishing a constant human presence on the Moon calls for developing new lunar human landing systems (HLS) which would deliver the crew from the prospective Lunar Gateway station to the surface of the Moon and back. Over the years, different human lunar lander architectures were proposed and multiple architecture studies were performed. However, those studies are relevant to the specific assumptions and lunar architectures proposed at the time of conducting the study. Since the current vision for lunar exploration includes new features, such as having the Lunar Gateway and switching to reusable systems, there is a need for a new HLS architecture study. Such studies are being performed by private companies; however, those are rarely publicly available. The goal of this paper is to address this gap and provide a publicly available architectural analysis within the current views on the future human lunar exploration.

We assume the Lunar Gateway in an L2 near rectilinear halo orbit and a landing site at the lunar South Pole; the number of HLS crew of 4; the surface stay time of ~ 7 days, the payload mass delivered to the surface of 500 kg, and the payload mass returned from the surface of 250 kg. A set of parametric models including an HLS model and an HLS program cost model is developed for the analysis. 39 architectures with varying number of stages (1, 2, and 3 stages) and propellant combinations (LOX/LH2, LOX/CH4, and MMH/NTO) are explored. The Pareto analysis shows that there is a difference between typical performance trends for expendable and reusable architectures. For expendable architectures, the 2-stage option seems to be the most advantageous while, for reusable architectures, the 1- and 3-stage options are either comparable or win over the 2-stage option even for the number of system uses as low as 3. In terms of the propellant combinations, pure LOX/LH2 or combined LOX/LH2/LOX/CH4 architectures dominate the tradespace. Assuming that the inter-stage propellant compatibility is a preferred option for systems refueling from the Gateway, 1-stage and 3-stage all LOX/LH2 architectures are identified as the likeliest candidates to have lowest HLS-related production and launch costs. Further cost analysis of those two architectures shows that the 1-stage HLS wins over the 3-stage system in terms of the overall HLS program cost if a long-term exploration program (on the order of tens of missions) is assumed.

Keywords: Moon exploration, human landing systems, space systems architecture, tradespace exploration.

1. Introduction

In 2017, the United States government has provided a clear direction for the US human spaceflight program for a return to the Moon by 2024 [1]. The focus of those efforts culminated in the development of the Artemis program [2], which forms the backbone of international collaboration for future human space exploration endeavors. The next steps for the US human spaceflight program, according to the policy at the time of writing of this paper, will likely result in the establishment of a deep space Gateway and sortie missions at the lunar South Pole. Reusable systems will likely be employed in such architecture, to ensure sustainable operations over the long term. Architectural decisions such as system reusability need to be evaluated in the context of all other system-level engineering decisions, in order to develop an integrated plan for technology development. In a recent paper, we proposed a technology roadmap for lunar human landing systems, which are key building blocks for future exploration of the Moon [3].

Over the years, different human landing system architectures have been considered; multiple architecture studies were conducted. Those studies, however, are relevant to the specific assumptions and lunar exploration architectures proposed at the time of conducting the study. Since today we have new assumptions (such as having the Lunar Gateway in the lunar architecture and switching to reusable systems), there is a need for a new human landing system architecture study. Such studies are being performed by private companies; however, those are not usually publicly available. Making well-informed system architecture decisions is important as they have a significant impact on the end outcomes of program cost and performance (as described in the well-known '80/20 rule' in the context of conceptual system design [4]).

The goal of this paper is to address the gap of an open architectural study of human landing systems (HLS) for future Moon exploration. We develop parametric mathematical models to enable a comprehensive tradespace exploration of future (potentially reusable) HLSs. Open-source parametric models allow for a publicly available architectural analysis, within the framework of future lunar exploration programs. The mathematical models proposed in this paper can be reused and adapted for a range of planetary exploration systems, and are thus of general interest to the space systems engineering community. We achieve our goal in two stages:

- First, we develop a set of figures of merit (FOMs) and a general HLS model applicable to each of the three architecture types (1-stage, 2-stage, or 3-stage). The HLS model is used to calculate the mass properties of a specific HLS which are then used to calculate the respective FOMs. Using those FOMs as preliminary proxies for the HLS program cost, we conduct a Pareto analysis on the overall HLS architecture tradespace and narrow down architecture options to a few promising candidates.
- Second, we develop an HLS program cost model. This includes developing a refueling vehicle model to model the costs associated with delivering the HLS elements and propellant from Earth to the Gateway (the launch costs). The HLS program cost model is then used to perform the cost analysis on the HLS candidates identified at the first stage. As a result, one of the architectures is selected as having the lowest costs under the assumptions of this study.

The remainder of this paper is structured as follows. Section 2 of this paper describes the methodology used in this study. Section 2.1 provides an overview of HLS architectures; section 2.2 lists the main study assumptions; sections 2.3 defines the figures of merit used in the study; sections 2.4 and 2.5 develop the HLS and refueling vehicle models, respectively; section 2.6 is devoted to the HLS program cost model. Section 3 presents the results of the Pareto (section 3.1.) and cost (section 3.2) analyses. Section 4 summarizes the findings of the study and identifies opportunities for future work.

2. Methodology

We adopt a comprehensive architecting approach to tradespace exploration in this paper. We have discussed our approach in previous papers [5, 6], including previous human spaceflight assessments [7, 8]. The methodology can be briefly described as follows. We first provide a brief overview of the system under investigation – human landing system. We then narrow the focus of the study with a set of assumptions, which also set the limitations of our investigations and point out avenues for future work. Finally, we develop a set of parametric models which form the basis for our further HLS architecture tradespace analysis.

2.1. Overview of HLS architectures

A lunar human landing system is a transportation system which purpose is to deliver humans (crew) to and from the lunar surface. In our previous paper, we have discussed the three key families of HLS architectures, according to the number of stages being employed [3]. We noted that while 1-stage, 2-stage and 3-stage architectures have been either considered in conceptual studies, or reached prototype stage, the only known HLS architecture tested on the Moon was the one of the Apollo Lunar Module (ALM), which was a 2-stage architecture. The ALM was a system able to accommodate 2 crew members. It was composed of a descent and an ascent modules, with the descent module left on the lunar surface after use. Other proposed HLS have been proposed for crew size up to 4, and surface stay time from 7 to 14 days. Another key architectural choice to consider is the choice of propellants, varying from hydrocarbon-derived proposals to liquid hydrogen. We consider all these architectural variables in this tradespace study, and evaluate their joint effect on all figures of merit of interest for performance and cost. In order to frame the study, we specify key study assumptions that are related to our envisioned concept of operations for future HLS.

2.2. Main study assumptions

We consider the following general assumptions in our HLS tradespace modeling:

- Three HLS architectures are considered including 1-stage, 2-stage, and 3-stage HLS.
- The number of crew is set to 4.
- The HLS delivers the crew from the Gateway in an L2 4:1 NRHO[1] to the lunar South Pole and then back to the Gateway.

[1] This orbit is close to the current NASA choice for the Gateway orbit which is an L2 9:2 NRHO.

- The surface stay time is ~ 7 days (which corresponds to one Gateway orbit period); assuming that the crew performs 1 EVA per day, this gives us 7 EVAs per lunar mission in total. The overall crew support time provided by the HLS (includes surface operations and transfers from/to the Gateway) is 10 days.
- Three main propellant combinations are considered for the HLS vehicles: NTO/MMH, LOX/CH4, and LOX/LH2; different stages of 2-stage and 3-stage HLS can use different propellants.
- The payload delivered by HLS to the lunar surface (in addition to the crew) is 500 kg; the payload returned by HLS from the surface is 250 kg.

These assumptions narrow the analysis to HLS focused on short-stay '*sortie*' missions. Such types of missions are assumed to take place through the first years of future human lunar exploration and during the following early stage of the lunar outpost (located at the lunar South Pole) initial operational capability. After these first milestones, the human landing system will be able to continue transporting people to and from the Moon but will need to undergo some modifications to be able to stay dormant at the outpost for longer periods of time, with a surface stay time up to 180 days.

Table 1 summarizes the assumptions outlined above in a morphological matrix. Combining architecture types with propellant combination options yields 39 HLS architectures.

Table 1. HLS morphological matrix.

Decision	**Options**		
Architecture type	1-stage	2-stage	3-stage
HLS stage propellant combination*	NTO/MMH	LOX/CH4	LOX/LH2
Gateway orbit	L2 4:1 NRHO		
Total crew support time	10 day		
Number of EVAs per mission	7		
Payload down mass, kg	500 kg		
Payload up mass, kg	250 kg		

* This decision is made separately for each of the HLS stages.

2.3. Figures of merit

A number of figures of merit (FOMs) serving as quantitative metrics for HLS performance and cost were used throughout the study. These FOMs include the HLS total dry and wet masses, the HLS average dry and wet masses per lunar mission, and the HLS reusability ratio.

2.3.1. The HLS total dry mass and average dry mass per lunar mission

The spacecraft dry mass is a typical FOM often used as a proxy for the system development and production costs when two or more spacecraft are compared to each other (within the same class, heavier vehicles typically tend to have higher development and production costs). We used two types of dry mass to compare the respective HLS-related costs: the HLS total dry mass and the HLS average dry mass per lunar mission.

We define the HLS total dry mass as the sum of the dry masses of the individual vehicles comprising the HLS. We assume that this mass is indicative of the HLS development costs, as well as the HLS production costs associated with the first mission for reusable systems or with each lunar mission for fully expendable systems. The HLS total dry mass is calculated as follows:

- For a 3-stage HLS, $m_{dry}^{HLS} = m_{dry}^{asc} + m_{dry}^{dsc} + m_{dry}^{tv}$.
- For a 2-stage HLS, $m_{dry}^{HLS} = m_{dry}^{asc} + m_{dry}^{dsc}$.
- For a 1-stage HLS, it is simply the dry mass of the single stage.

In the formulas above, m_{dry}^{asc}, m_{dry}^{dsc}, and m_{dry}^{tv} are the dry masses of the ascent vehicle, descent vehicle and transfer vehicle, respectively.

We define the HLS average dry mass per mission as the sum of the dry masses of all HLS vehicles used per a given number of lunar missions (equal to the number of HLS uses n_{uses}) divided by the number of missions. We assume that this FOM is indicative of the average HLS production costs per one lunar mission. The HLS average dry mass per mission is calculated as follows:

- For a 3-stage HLS, $m_{dry}^{HLS\,\mathrm{ave}} = \left(m_{dry}^{asc} + m_{dry}^{tv} + n_{uses} m_{dry}^{dsc}\right) / n_{uses}$.
- For a 2-stage HLS, $m_{dry}^{HLS\,\mathrm{ave}} = \left(m_{dry}^{asc} + n_{uses} m_{dry}^{dsc}\right) / n_{uses}$.

- For a 1-stage HLS, $m_{dry}^{HLS\,\mathrm{ave}} = m_{dry}^{HLS} / n_{uses}$.

One notices that, for expendable HLS, the HLS average dry mass per mission is equal to the HLS total dry mass whereas, for reusable systems, the average dry mass is lower than the total dry mass as some parts of the HLS are reused and, therefore, are not produced for the subsequent missions (hence, the production costs for such systems are expected to be, on average, lower).

2.3.2. The HLS total wet mass and average wet mass per lunar mission

The spacecraft wet mass is another FOM which is traditionally used to compare spacecraft and which can serve as a proxy for the system launch costs – i.e., the costs associated with delivering the system from the surface of the Earth to its destination in space. The obvious rule here is that the heavier the vehicle, the higher its launch costs. Similarly to dry mass, we used two types of wet mass to compare the HLS-related launch costs: the HLS total wet mass and the HLS average wet mass per lunar mission.

We define the HLS total wet mass as the sum of the wet masses of the individual vehicles comprising the HLS and assume that this mass is indicative of the HLS-related launch costs associated with the first mission for reusable systems or with each lunar mission for fully expendable systems. The HLS total wet mass is calculated as follows:

- For a 3-stage HLS, $m_{wet}^{HLS} = m_{wet}^{asc} + m_{wet}^{dsc} + m_{wet}^{tv}$.
- For a 2-stage HLS, $m_{wet}^{HLS} = m_{wet}^{asc} + m_{wet}^{dsc}$.
- For a 1-stage HLS, it is simply the wet mass of the single stage.

In the formulas above, m_{wet}^{asc}, m_{wet}^{dsc}, and m_{wet}^{tv} are the wet masses of the ascent vehicle, descent vehicle and transfer vehicle, respectively.

We define the HLS average wet mass per mission as the sum of the wet masses of all HLS vehicles used per a given number of lunar missions (equal to the number of HLS uses n_{uses}) divided by the number of missions. We assume that this FOM is indicative of the average HLS-related launch costs per one lunar mission. The HLS average wet mass per mission is calculated as follows:

- For a 3-stage HLS, $m_{wet}^{HLS\,\mathrm{ave}} = \left(m_{dry}^{asc} + m_{dry}^{tv} + n_{uses}\left(m_{dry}^{dsc} + m_{p}^{asc} + m_{p}^{dsc} + m_{p}^{tv}\right)\right) / n_{uses}$.
- For a 2-stage HLS, $m_{wet}^{HLS\,\mathrm{ave}} = \left(m_{dry}^{asc} + n_{uses}\left(m_{dry}^{dsc} + m_{p}^{asc} + m_{p}^{dsc}\right)\right) / n_{uses}$.
- For a 1-stage HLS, $m_{wet}^{HLS\,\mathrm{ave}} = \left(m_{dry}^{HLS} + n_{uses} m_{p}^{HLS}\right) / n_{uses}$.

In the formulas above, m_{p}^{asc}, m_{p}^{dsc}, and m_{p}^{tv} are the propellant masses of the ascent vehicle, descent vehicle and transfer vehicle, respectively; m_{p}^{HLS} is the HLS total propellant mass.

One notices that for expendable HLS, the HLS average wet mass per mission is equal to the HLS total wet mass whereas, for reusable systems, the average wet mass is lower than the HLS total wet mass as some parts of the system are reused and do not need to be launched for subsequent missions since they are already in space after their first use (hence, the launch costs for reusable systems are expected to be, on average, lower).

2.3.3. Reusability ratio

To estimate '*the degree of reusability*' of the system, we use the HLS reusability ratio which we define as the total dry mass of all reusable HLS vehicles/parts divided by the total dry mass of all HLS vehicles/parts (reusable and expendable):

$$k_r = \frac{\sum m_{dry\,i}^{\mathrm{reusable}}}{\sum m_{dry\,i}^{\mathrm{reusable}} + \sum m_{dry\,i}^{\mathrm{expendable}}} = \frac{\sum m_{dry\,i}^{\mathrm{reusable}}}{m_{dry}^{HLS}}$$

This ratio shows how much of the system's dry mass is reused in the subsequent missions: the zero value means that the HLS is fully expendable; the value of 1 (or 100%) means that the HLS is fully reusable. The reusability ratio allows relating the HLS average dry mass per mission to the HLS total dry mass via a simple expression holding for all HLS architectures:

$$m_{dry}^{HLS\,\mathrm{ave}} = m_{dry}^{HLS}\left(1 - k_r \frac{n_{uses} - 1}{n_{uses}}\right)$$

2.4. Human landing system model

Given an architecture type (1-stage, 2-stage, or 3-stage) and the propellant combinations for the respective stages, the HLS model provides HLS sizing equations solving which one can obtain mass properties for a specific HLS. For each architecture type, HLS sizing equations are developed based on the respective concept of operations and mass estimating relationships for the individual HLS vehicles (ascent stage, descent stage, and transfer vehicle).

2.4.1. Concept of operations and delta-V requirements

It is assumed that the HLS operates within the current NASA lunar exploration framework and delivers crew from the Lunar Gateway station in a near rectilinear halo orbit (NRHO) to the lunar surface and from the lunar surface back to the station [9]. The concept of operations for each of the staging options considered is described below (see Fig. 1).

Three-stage architecture includes three vehicles – ascent stage, descent stage, and transfer vehicle. The respective concept of operations can be described as follows:

- the stack consisting of the ascent stage, descent stage and transfer vehicle starts from the Gateway orbit (the crew is in the ascent stage);
- the transfer vehicle transfers the stack to a low lunar orbit, separates from the stack, and returns back to the Gateway;
- the descent stage lands itself and the ascent stage on the lunar surface;
- after the surface operations are over, the ascent stage with the crew and collected lunar samples ascends to a low lunar orbit leaving the descent stage on the Moon;
- the ascent stage then returns from the low lunar orbit to the Gateway.

Under this architecture the transfer vehicle and ascent stage are deemed reusable, while the descent stage is considered expendable.

Two-stage architecture includes two vehicles – ascent stage and descent stage – and considers the following concept of operations:

- the stack consisting of the ascent and descent stages starts from the Gateway orbit (the crew is in the ascent stage);
- the descent stage transfers the stack from the Gateway orbit to a low lunar orbit, then performs the descent from the lunar orbit to the lunar surface;
- after the surface operations are over, the ascent stage with the crew and collected lunar samples ascends to a low lunar orbit leaving the descent stage on the Moon;
- the ascent stage then returns from the lunar orbit to the Gateway.

Under this architecture, the ascent stage is reusable; the descent stage is expendable.

One-stage architecture includes only one vehicle and implements the following concept of operations:

- starting from the Gateway orbit, the HLS with the crew onboard transfers itself to a low lunar orbit and then descends from this orbit to the lunar surface;
- after the surface operations are over, the HLS with the crew and samples onboard ascends to a low lunar orbit and then returns back to the Gateway.

In this architecture case, the HLS is fully reusable.

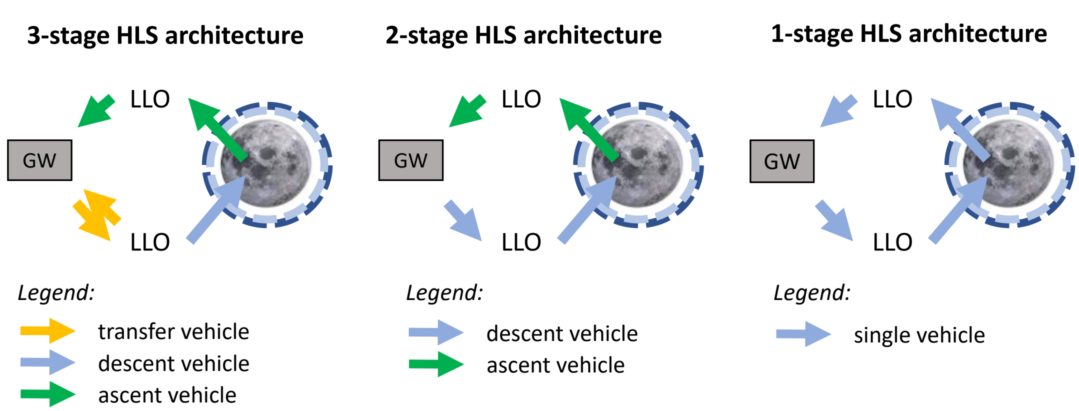

Fig. 1. HLS concepts of operation (in the figure, GW stands for Gateway, LLO for low lunar orbit).

The delta-V requirements for the Gateway in an L2 4:1 NRHO [10] and their allocation to the individual HLS vehicles are presented in Table 2.

Table 2. Delta-V allocation to individual HLS vehicles.

Operation	Notation	Delta-V, m/s	Function assignment		
			1-stage HLS	2-stage HLS	3-stage HLS
Transfer from Gateway orbit to LLO	ΔV_{GW-LLO}	780	single stage	descent stage	transfer vehicle
Transfer from LLO to lunar surface	ΔV_{LLO-LS}	1,900	single stage	descent stage	descent stage
Transfer from lunar surface to LLO	ΔV_{LS-LLO}	1,900	single stage	ascent stage	ascent stage
Transfer from LLO to Gateway orbit (crew)	ΔV_{LLO-GW}^{crew}	820	single stage	ascent stage	ascent stage
Transfer from LLO to Gateway orbit (transfer vehicle)	ΔV_{LLO-GW}^{tv}	740	-	-	transfer vehicle

* LLO stands for low lunar orbit. LLO is assumed to be a polar circular 100-km orbit.

2.4.2. Mass estimating relationships for individual HLS vehicles

Mass estimating relationships from [11, 12] were selected for modeling individual HLS vehicles for their simplicity and suitability for modeling all three vehicles.

The ascent vehicle consists of a crew cabin and a propulsion module. It also carries consumables needed for the mission. The respective mass components are estimated as follows:

$$\begin{gathered} m_{cabin} = 1,250 + 525 n_{crew} \\ m_{consumables} = 9.4 n_{crew} t_{support} + 2.3 t_{support} + 4.5 n_{EVA cycles} \\ m_{dry\,PM}^{asc} = A_a m_{launch} + B_a m_p^{asc} + C_a \\ m_{dry}^{asc} = m_{cabin} + m_{dry\,PM}^{asc}, \qquad m_{wet}^{asc} = m_{dry}^{asc} + m_{consumables} + m_p^{asc} \end{gathered} \tag{1}$$

Here, n_{crew} is the number of crew; $t_{support}$ is the total number of days the crew cabin provides life support to the crew (including in-space transportation and surface stay); $n_{EVA cycles}$ is the number of EVA cycles to be performed on the surface; A_a, B_a, C_a are the model coefficients for the ascent vehicle; m_{launch} is the total wet mass being launched from the lunar surface (this mass depends on the HLS architecture type); m_p^{asc} is the ascent vehicle propellant mass; $m_{dry\,PM}^{asc}$, m_{dry}^{asc}, and m_{wet}^{asc} are the dry mass of the ascent propulsion module, the ascent vehicle total dry mass, and the ascent vehicle wet mass respectively (all masses are in kg).

The descent vehicle dry and wet masses are calculated as follows:

$$m_{dry}^{dsc} = A_d m_{deorbit} + B_d m_p^{dsc} + C_d, \qquad m_{wet}^{dsc} = m_{dry}^{dsc} + m_p^{dsc} \tag{2}$$

Here, A_d, B_d, C_d are the model coefficients for the descent vehicle; $m_{deorbit}$ is the total wet mass being de-orbited prior to descent to the lunar surface (this mass depends on the HLS architecture type); m_p^{dsc} is the propellant mass of the descent vehicle.

The model coefficients are defined as follows:

$$\begin{gathered} A_a = A_d = 0.0640, \qquad C_a = C_d = 390 \text{ kg} \\ B_a = 0.0506 \rho_{bulk}^{ALM} / \rho_{bulk}^{asc}, \quad B_d = 0.0506 \rho_{bulk}^{ALM} / \rho_{bulk}^{dsc} \end{gathered}$$

Here, ρ_{bulk}^{ALM} is the bulk density of the Apollo Lunar Module propellant (A-50/NTO); ρ_{bulk}^{asc} and ρ_{bulk}^{dsc} are the propellant bulk densities of the ascent and descent vehicles respectively. The propellant bulk density for a specific type of propellant is calculated as follows:

$$\rho_{bulk} = \left(r_p + 1\right) / \left(r_p / \rho_{ox} + 1 / \rho_{fu}\right)$$

Here r_p is the oxidizer-to-fuel ratio, ρ_{ox} is the oxidizer density, and ρ_{fu} is the fuel density. Table 3 contains the propellant characteristics used for calculations.

Table 3. Propellant characteristics and transfer vehicle model coefficients used for calculations.

Propellant type	ρ_{ox}**, kg/m**^{3}	ρ_{fu}**, kg/m**^{3}	r_p	I_{sp}**, s**	B_t	C_t**, kg**
LOX/LH2	1,141	71	5.88	450	0.04545	2,279
LOX/LCH4	1,141	424	3.60	363	0.04545	2,279
MMH/NTO	1,442	870	1.90	320	0.04253	2,454
A-50/NTO (ALM)	1,434	897	1.60	311	-	-

The transfer vehicle dry and wet masses are calculated as follows:

$$m_{dry}^{tv} = B_t m_p^{tv} + C_t, \qquad m_{wet}^{tv} = m_{dry}^{tv} + m_p^{tv}$$

Here m_p^{tv} is the propellant mass of the transfer vehicle; B_t and C_t are the model coefficients (see Table 3).

2.4.3. HLS Sizing Equations

For each architecture type, the masses m_{launch} and $m_{deorbit}$ were defined through the mass components of the individual HLS vehicles according to the respective concepts of operations. The resultant mass equations were complemented with the rocket equation expressions for the delta-Vs allocated to the vehicles, expressed in the following linear form:

$$m_{fi} - E_i m_{0i} = 0, \qquad E_i = \exp\left(-\Delta V_i / [I_{sp\,i} g_0]\right)$$

Here, m_{0i}, m_{fi} are the masses of the respective vehicle stack before and after the propulsive maneuver; $I_{sp\,i}$ is the specific impulse of the vehicle performing the burn; $g_0 = 9.81\,\text{m/s}^2$ is the Earth's standard gravitational acceleration.

As a result, for each staging option, a system of linear equations with HLS mass components as unknowns was produced. The 3-stage HLS model combined the models of all three individual vehicles (ascent, descent, and transfer vehicles) and was characterized by the following system:

$$\begin{pmatrix} 1-E_a & -E_a & 0 & 0 & 0 & 0 & 0 \\ A_a - 1 & A_a + B_a & 0 & 0 & 0 & 0 & 0 \\ 1-E_d & 1-E_d & 1-E_d & -E_d & 0 & 0 & 0 \\ A_d & A_d & A_d - 1 & A_d + B_d & 0 & 0 & 0 \\ 1-E_{t1} & 1-E_{t1} & 1-E_{t1} & 1-E_{t1} & 1-E_{t1} & -E_{t1} & 1-E_{t1} \\ 0 & 0 & 0 & 0 & 1-E_{t2} & 0 & -E_{t2} \\ 0 & 0 & 0 & 0 & -1 & B_t & B_t \end{pmatrix} \begin{pmatrix} m_{dry\,PM}^{asc} \\ m_p^{asc} \\ m_{dry}^{dsc} \\ m_p^{dsc} \\ m_{dry}^{tv} \\ m_{p1}^{tv} \\ m_{p2}^{tv} \end{pmatrix} = \begin{pmatrix} -\left(1-E_a\right) m_{up} \\ -A_a m_{up} - C_a \\ -\left(1-E_d\right) m_{down} \\ -A_d m_{down} - C_d \\ -\left(1-E_{t1}\right) m_{down} \\ 0 \\ -C_t \end{pmatrix} \tag{3}$$

$$E_{t1} = \exp\left(-\Delta V_{GW-LLO} / \left[I_{sp}^{tv} g_0\right]\right), \qquad E_{t2} = \exp\left(-\Delta V_{LLO-GW}^{tv} / \left[I_{sp}^{tv} g_0\right]\right),$$

$$E_d = \exp\left(-\Delta V_{LLO-LS} / \left[I_{sp}^{dsc} g_0\right]\right), \; E_a = \exp\left(-\left[\Delta V_{LS-LLO} + \Delta V_{LLO-GW}^{crew}\right] / \left[I_{sp}^{asc} g_0\right]\right)$$

$$m_{p1}^{tv} + m_{p2}^{tv} = m_p^{tv}$$

The system for the 2-stage HLS model combined the models of the descent and ascent vehicles:

$$\begin{pmatrix} 1-E_a & -E_a & 0 & 0 & 0 \\ A_a-1 & A_a+B_a & 0 & 0 & 0 \\ 1-E_{d1} & 1-E_{d1} & 1-E_{d1} & -E_{d1} & 1-E_{d1} \\ A_d & A_d & A_d-1 & B_d & A_d+B_d \\ 1-E_{d2} & 1-E_{d2} & 1-E_{d2} & 0 & -E_{d2} \end{pmatrix} \begin{pmatrix} m_{dry\,PM}^{asc} \\ m_p^{asc} \\ m_{dry}^{dsc} \\ m_{p1}^{dsc} \\ m_{p2}^{dsc} \end{pmatrix} = \begin{pmatrix} -(1-E_a)m_{up} \\ -A_a m_{up} - C_a \\ -(1-E_{d1})m_{down} \\ -A_d m_{down} - C_d \\ -(1-E_{d2})m_{down} \end{pmatrix}$$

$$E_{d1} = \exp\left(-\Delta V_{GW-LLO} / \left[I_{sp}^{dsc} g_0\right]\right), \qquad E_{d2} = \exp\left(-\Delta V_{LLO-LS} / \left[I_{sp}^{dsc} g_0\right]\right) \tag{4}$$
$$E_a = \exp\left(-\left[\Delta V_{LS-LLO} + \Delta V_{LLO-GW}^{crew}\right] / \left[I_{sp}^{asc} g_0\right]\right), \qquad m_{p1}^{dsc} + m_{p2}^{dsc} = m_p^{dsc}$$

The 1-stage HLS model was built as a mixed combination of the ascent and descent vehicle models. Similarly to the ascent vehicle, the single-stage HLS has a crew cabin and carries consumables which masses are defined by the respective equations from Eqs. (1). However, a more demanding Eq. (2) was used to relate the vehicle's propulsion module dry mass to the de-orbit and propellant masses. The resultant sizing linear system is as follows (the superscript 'ss' in the equations below stands for 'single stage'):

$$\begin{pmatrix} 1-E_1 & -E_1 & 1-E_1 \\ A_d-1 & B_d & A_d+B_d \\ 1-E_2 & 0 & -E_2 \end{pmatrix} \begin{pmatrix} m_{dry\,PM}^{ss} \\ m_{p1}^{ss} \\ m_{p2}^{ss} \end{pmatrix} = \begin{pmatrix} -(1-E_1)m_{down} \\ -A_d m_{down} - C_d \\ -(1-E_2)m_{down} \end{pmatrix}$$

$$E_1 = \exp\left(-\Delta V_{GW-LLO} / \left[I_{sp}^{ss} g_0\right]\right), \quad E_2 = \exp\left(-\left[\Delta V_{LLO-LS} + \Delta V_{LS-LLO} + \Delta V_{LLO-GW}^{crew}\right] / \left[I_{sp}^{ss} g_0\right]\right) \tag{5}$$
$$m_{p1}^{ss} + m_{p2}^{ss} = m_p^{ss}, \qquad m_{dry}^{ss} = m_{cabin} + m_{dry\,PM}^{ss}, \qquad m_{wet}^{ss} = m_{dry}^{ss} + m_{consumables} + m_p^{ss}$$

In Eqs. (3)-(5), parameters m_{down} and m_{up} are defined as follows:

$$m_{down} = m_{cabin} + m_{consumables} + m_{PLD}^{down}, \qquad m_{up} = m_{cabin} + m_{consumables} + m_{PLD}^{up}$$

Here, m_{PLD}^{down}, m_{PLD}^{up} are the masses of the payload delivered to and from the lunar surface, respectively.

2.5. Refueling vehicle model

A simple refueling vehicle (RV) model was used to model HLS delivery from LEO to the Gateway in order to estimate the HLS-related launch costs; we developed a parametric model for RV operations using a similar approach to previous work on launch vehicle upper stage refueling operations [13]. It was assumed that the refueling vehicle (RV) delivers HLS propellant for lunar missions from LEO to the Gateway. If partially fueled, it can also deliver HLS vehicles to the station.

2.5.1. Concept of operations

The following concept of operations for HLS delivery is assumed:

- The RV is launched to LEO on top of the Falcon Heavy launch vehicle; if HLS vehicle(s) are part of the cargo, they are attached to RV and launched together with it;
- Once in LEO, RV uses its own propulsion to get to the station performing the trans-lunar injection and Gateway orbit insertion maneuvers;
- After the cargo is transferred to the station, RV undocks and is disposed of.

For the first mission in the HLS life cycle, all individual HLS vehicles, their propellant and consumables are delivered by the refueling vehicle to the Gateway; for each of the subsequent missions, the RV delivers the new expendable descent stage (for 2-stage and 3-stage architectures), propellant for all HLS stages, and a new load of consumables. For calculation purposes, it is assumed that the payload that the HLS delivers to the lunar surface is delivered to the Gateway separately (for example, with the Gateway resupply missions). That is, the down-payload mass is used for HLS sizing but is not used for HLS-related launch costs estimates as it is not really part of the HLS.

2.5.2. Model assumptions and RV mass properties derivation

It was assumed that the refueling vehicle uses LOX/LH2 propulsion with a specific impulse of 450 s. The maximum payload mass Falcon Heavy delivers to LEO is 63,800 kg which gives the RV wet mass.

An approach adapted from [13] was used to derive the propellant mass and the dry mass of the refueling vehicle from its wet mass. The refueling vehicle is assumed to be similar to a traditional LOX/LH2 upper stage but with refueling equipment added to it: the additional equipment allows transfer of propellant from the RV tanks to the Gateway storage facility. RV uses the same tanks for propellant transportation and its own propulsion needs. It was assumed that the mass fraction (defined as the ratio of the vehicle's dry mass to its wet mass) the RV would have without the refueling equipment is $\mu_0 = 0.1$; the refueling equipment adds 15% to the dry mass of RV. The following formulas for the RV maximum propellant load and dry mass were derived based on those assumptions:

$$m_{p\max}^{RV} = \frac{1-\mu_0}{1+\mu_0\lambda} m_{wet}^{RV}, \qquad m_{dry}^{RV} = \frac{\mu_0\left(1+\lambda\right)}{1-\mu_0} m_{p\max}^{RV}$$

Plugging in $\mu_0 = 0.1$, $\lambda = 0.15$, and $m_{wet}^{RV} = 63{,}800\text{ kg}$, one gets $m_{p\max}^{RV} = 56{,}571\text{ kg}$. This is the maximum propellant load for which the RV tanks are designed. The RV dry mass is $m_{dry}^{RV} = 7{,}229\text{ kg}$.

The RV uses part of its propellant on the trans-lunar injection and Gateway orbit insertion maneuvers. One can use the rocket equation to estimate the maximum payload mass (in form of propellant and HLS vehicles) delivered by the RV to the Gateway:

$$m_{PLDtoGW}^{RV} = E m_{wet}^{RV} - m_{dry}^{RV}, \qquad E = \exp\left(-\frac{\Delta V_{TLI} + \Delta V_{GWOI}}{I_{sp}^{RV} g_0}\right)$$

Assuming $\Delta V_{TLI} = 3{,}140\text{ m/s}$ and $\Delta V_{GWOI} = 430\text{ m/s}$ [10], one gets $m_{PLDtoGW}^{RV} = 21{,}190\text{ kg}$.

2.6. HLS program cost model

To compare different HLS architectures in terms of the respective HLS-related mission costs, the total HLS program cost model was developed based on the cost estimating relationships from [14] (all model coefficients were adjusted to 2020 for inflation). The total HLS program cost was estimated as the sum of the HLS development cost, production costs of all HLS vehicles to be used during the lunar program, and costs of delivery of all HLS-related vehicles, propellant, and consumables from the surface of the Earth to the Gateway station:

$$C_{program}^{HLS} = C_{development}^{HLS} + C_{production\,\Sigma}^{HLS} + C_{launch\,\Sigma}^{HLS}$$

The respective cost components are calculated as follows:

$$C_{development}^{HLS} = c_{dev}^{HLS} m_{dry}^{HLS}, \qquad C_{production\,\Sigma}^{HLS} = c_{prod}^{HLS} M_{dry\,\Sigma}, \qquad C_{launch\,\Sigma}^{HLS} = C_{program}^{RV}$$

Here, $M_{dry\,\Sigma}$ is the total dry mass of all HLS vehicles to be produced during the lunar program; $C_{program}^{RV}$ is the total cost of the refueling vehicle program; $c_{dev}^{HLS} = 832{,}000\text{ USD/kg}$, $c_{prod}^{HLS} = 65{,}000\text{ USD/kg}$ are the model coefficients.

The total cost of the refueling vehicle program is estimated as the sum of the RV development cost, production costs of all refueling vehicles to be used during the lunar program and the costs of their launch to LEO (by the Falcon Heavy launch vehicle):

$$C_{program}^{RV} = C_{development}^{RV} + C_{production\,\Sigma}^{RV} + C_{launch\,\Sigma}^{RV}$$
$$C_{development}^{RV} = c_{dev}^{RV} m_{dry}^{RV}, \qquad C_{production\,\Sigma}^{RV} = c_{prod}^{RV} n_{RV} m_{dry}^{RV}, \qquad C_{launch\,\Sigma}^{RV} = n_{RV} C_{Falcon\,Heavy\,launch}$$

Here, n_{RV} is the total number of refueling vehicles to be produced during the lunar program; $C_{Falcon\,Heavy\,launch} = 150\ \$M$ is the Falcon heavy launch cost for the 63,800-kg payload to LEO [15]; $c_{dev}^{RV} = 207{,}000\ \text{USD/kg}$, $c_{prod}^{RV} = 14{,}000\ \text{USD/kg}$ are the model coefficients.

The total number of refueling vehicles needed for the lunar program can be estimated as follows:

$$n_{RV} = M_{toGW\,\Sigma} / m_{PLDtoGW}^{RV}$$

Here, $M_{toGW\,\Sigma}$ is the total mass of all HLS-related vehicles, propellant and consumables to be delivered to the Gateway during the lunar program.

The HLS total mass sums used in the program models are calculated as follows:

$$M_{dry\,\Sigma} = n_{HLS\,cycles} m_{dry}^{HLS} \left(1 + \left(n_{uses} - 1\right)\left(1 - k_r\right)\right)$$

$$M_{prop\&cons} = n_{missions} \left(m_p^{HLS} + m_{consumables}\right), \qquad M_{toGW\,\Sigma} = M_{dry\,\Sigma} + M_{prop\&cons}$$

The total number of missions $n_{missions}$ and the number of reusable landing systems $n_{HLS\,cycles}$ required per lunar program are as follows:

$$n_{missions} = T_{program} n_{missions\,per\,year}, \qquad n_{HLS\,cycles} = n_{missions} / n_{uses}$$

Here, $T_{program}$ is the lunar program duration (in years); $n_{missions\,per\,year}$ is the number of lunar missions per year. Knowing the total HLS program cost, one can estimate the average HLS-related mission costs:

$$C_{mission}^{HLS} = C_{program}^{HLS} / n_{missions}$$

3. Results

3.1. Pareto analysis

HLS architectures with different numbers of stages (1, 2, or 3 stages) and different types of propellant for each of the stages (LOX/LH2, LOX/CH4, or MMH/NTO) were explored. Pareto charts for different numbers of HLS system uses are shown in Fig. 2. The two axes of the upper-left chart are the total HLS wet mass in the Gateway orbit (the horizontal axis) and the total HLS dry mass (the vertical axis). Those masses are used here as proxies for the launch and production costs associated either with every mission in case of expendable systems (number of system uses is 1) or with the first mission of the HLS lifecycle in case of reusable systems. In order to analyze the effect of reusability on different staging options, the remaining three charts use the average HLS wet mass and the average HLS dry mass delivered to the Gateway orbit (GWO) per mission as their axes. The average values serve as proxies for the respective average costs. The charts are plotted for the cases of 3, 5, and 10 uses. The utopia point for all four charts is the bottom-left corner. The charts allow identifying systems which are likely to have lowest total costs without making any assumptions on how the production and launch costs compare to each other at this point: the Pareto-dominant systems will have both their dry and wet masses – and, presumably, the production and launch costs – lower than the Pareto-dominated ones.

One observes that, in case of fully expendable systems (see the upper-left Pareto chart in Fig. 2), 2-stage architectures seem to be the most advantageous: having both lower total dry masses and propellant loads, they form most of the respective Pareto frontier and will presumably produce lower production and launch costs per mission.

The situation is different, however, for reusable systems. Even the most effective 2-stage architectures have the lowest reusability ratio (about 60% compared to 70-75% and 100% for the 3-stage and 1-stage architectures respectively). As a result, 3-stage architectures become comparable to 2-stage architectures in terms of average dry mass and average total mass to be delivered to the Gateway per mission (and, thus, presumably comparable in the average mission costs) even for systems designed only for 3 uses (see the upper-right Pareto chart in Fig. 2). Further increase in the number of uses moves 3-stage architectures forward on the Pareto frontier in front of the 2-stage architectures. At the same time, 1-stage architectures benefit from reusability even more, forming the lower right part of the Pareto frontier starting with the case of 3 uses. Those architectures are characterized by considerably lower average dry mass per mission compared to the Pareto-frontier 2-stage and 3-stage architectures while losing to those architectures in average total wet mass per mission.

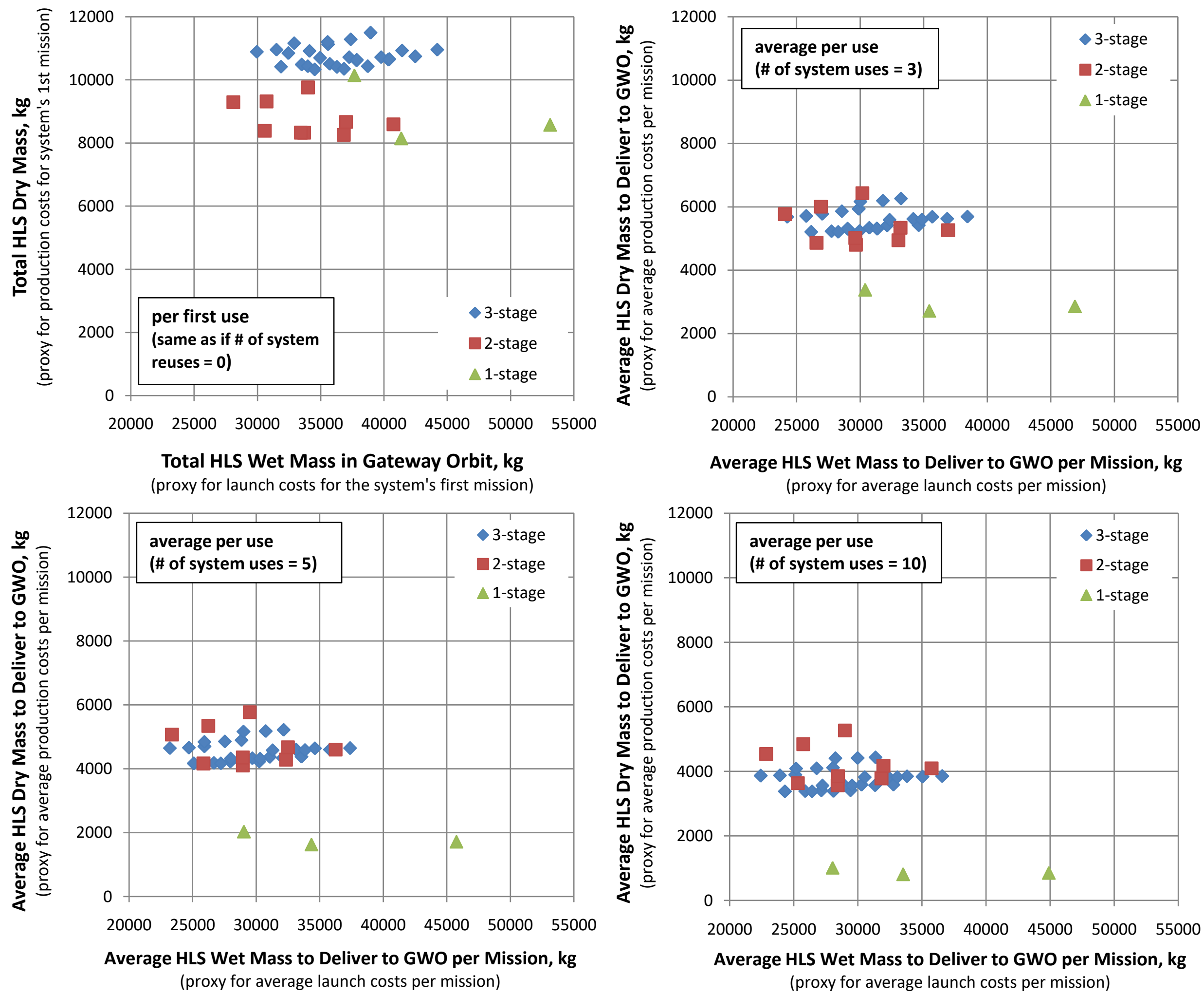

Fig. 2. Pareto charts for different numbers of system uses.

Table 4. Architectures on the Pareto frontier in case of 10 system uses.

Arch #	# of stages	Average HLS wet mass per mission, kg [~ launch costs]	Average HLS dry mass per mission, kg [~ production costs]	HLS total dry mass, kg [~ dev'ment costs]	HLS total wet mass*, kg	Reusa-bility Ratio	Stage details (dry mass, kg/ propellant mass, kg/ propellant type)
3-1	3	22,429	3,865	10,892	29,956	0.72	ascent stage: 5,287/5,083/LOX/LH2 descent stage: 3,085/7,738/LOX/LH2 transfer vehicle: 2,520/5,312/LOX/LH2
3-4	3	24,318	3,382	10,422	31,857	0.75	ascent stage: 5,287/5,083/LOX/LH2 descent stage: 2,600/9,800/LOX/CH4 transfer vehicle: 2,535/5,622/LOX/LH2
1-1	1	28,025	1,015	10,152	37,662	1.00	single-stage HLS: 10,152/26,579/LOX/LH2
1-2	1	33,531	815	8,149	41,365	1.00	single-stage HLS: 8,149/32,285/LOX/LCH4

* includes the mass of consumables and the down-payload mass

Gains in the average dry mass and average wet mass per mission due to the increase in the number of uses tend to decrease with each additional use added: while there is a visible difference between Pareto charts for 1, 3, and 5 uses, charts for 5 and 10 uses (the bottom-left and bottom-right charts in Fig. 2) look rather similar. For further analysis, it was assumed that the HLS is developed for 10 uses. Table 4 contains details on the Pareto frontier points for this case.

Of the four architectures on the Pareto frontier (see Table 4), two are pure LOX/LH2 architectures, one uses LOX/LH2 for its ascent and transfer stages and LOX/CH4 for its descent stage, and the last architecture (a single-stage HLS) uses only LOX/CH4. None of the architectures uses NTO/MMH. Considering the fact that having the same propellant components for all HLS elements might be preferable (due to compatibility issues and for more effective refueling operations), it seems reasonable to prioritize the development of the LOX/LH2 propulsion as the main propulsion technology for the HLS. The pure LOX/LH2 architectures 3-1 and 1-1 were selected for further analysis.

3.2. Cost Analysis

The following lunar program assumptions were used to estimate the HLS program cost: the program duration is at least 20 years; the number of lunar missions per year is two. This gives the total number of lunar missions of 40. Based on the assumptions stated here and in the previous sections, the HLS program cost components and the expected program and mission costs were calculated for the two architectures selected for further investigation. The results are presented in Table 5.

Table 5. HLS program cost for a 40-mission lunar program (in 2020 USD).

Mass/Cost component	**LOX/LH2 architecture**	
	3-stage	**1-stage**
HLS total dry mass, kg	10,892	10,152
HLS propellant mass, kg	18,133	26,579
HLS consumables mass, kg	430.5	430.5
Reusability ratio	0.72	1.00
HLS Mass Sums per Program		
total dry mass of all HLS vehicles, kg	153,359	40,608
total mass of propellant and consumables, kg	742,540	1,080,380
total mass to deliver to Gateway, kg	895,899	1,120,988
Refueling Vehicle Program Cost, mln USD	**12,298**	**14,810**
RV dry mass, kg	7,229	7,229
payload mass delivered by RV to Gateway, kg	21,190	21,190
number of RVs needed per program	43	53
RV development cost, mln USD	1,496	1,496
RV production costs, mln USD	4,352	5,364
RV launch costs, mln USD	6,450	7,950
launch cost per kg delivered to Gateway, USD/kg	13,727	13,212
Human Landing System Program Cost, mln USD	**31,329**	**25,896**
HLS development cost, mln USD	9,062	8,446
HLS production costs, mln USD	9,968	2,640
HLS launch costs, mln USD	12,298	14,810
HLS-related costs per mission, mln USD	**783**	**647**

One observes that program cost for the 1-stage HLS is 21% lower than for the 3-stage HLS. In order to estimate the sensitivity of the obtained results to the program assumptions, the cost calculations were repeated for different values of the number of missions per lunar program in the range from 10 to 100 missions. The results are shown in Fig. 3. One observes that the dominance of the 1-stage architecture over the 3-stage one in terms of the program and average mission costs holds over the whole range. The difference in cost further increases with the increase in the number of missions. The current NASA's intention to establish constant human presence on the Moon suggests that the number of lunar missions is likely to be way more than 10 which, under the cost assumptions used here, makes the 1-stage architecture a preferred choice.

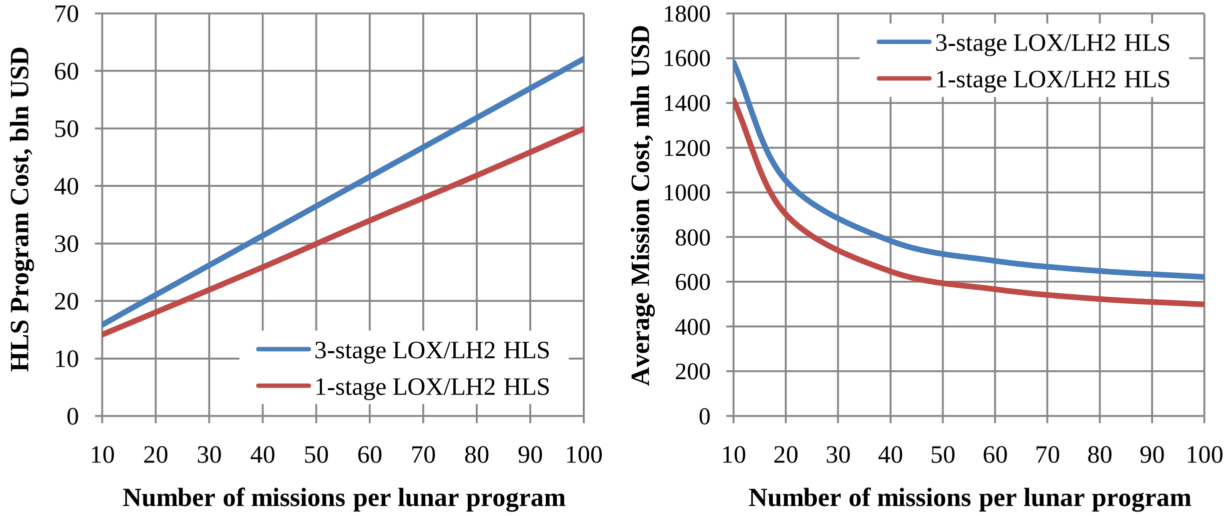

Fig. 3. Sensitivity of the HLS-related program and average mission costs to the number of missions.

The observed cost advantage of the 1-stage HLS can be explained in the following way. Due to its higher reusability ratio, the 1-stage HLS requires fewer HLS vehicles to be produced (~ 41 t of total dry mass compared to 153 t in case of the 3-stage option). At the same time, it requires more refueling vehicles to be produced and launched (53 RVs in total compared to 43 RVs in case of the 3-stage option). However, since human-rated HLS vehicles are more complex than the refueling vehicle, in terms of the overall cost, the decrease in the number of HLS elements outweighs the increase in the number of RVs: it is cheaper to produce and launch more refueling vehicles than to produce more HLS elements. The respective differences in the numbers of HLS elements and RVs between the two architectures increase even further with the increase in the total number of missions thus making the 1-stage HLS even more attractive in case of a more extensive lunar program.

4. Conclusion

In this study, we performed a human landing system architecture tradespace analysis relevant to the current vision of lunar exploration which includes the Lunar Gateway in an L2 near rectilinear halo orbit and tasks a reusable HLS with delivering the crew from the Gateway to a landing site at the lunar South Pole and back to the Gateway. A set of parametric models including an HLS model applicable to the 1-, 2-, and 3-stage cases and an HLS program cost model were developed for the analysis.

Thirty-nine different HLS architectures with varying number of stages and propellant combinations were explored. The analysis of the Pareto charts plotted for different numbers of HLS uses showed that it yields different results for expendable and reusable systems. In case of fully expendable HLS, 2-stage architectures proved to be the most advantageous: having both lower total dry masses and propellant loads, they form most of the respective Pareto frontier and presumably produce lower production and launch costs per mission. However, when reusability is introduced, 3-stage architectures (which have a higher reusability ratio than the 2-stage systems) become comparable in their performance to 2-stage architectures even for systems designed only for 3 uses. Further increase in the number of uses moves 3-stage architectures forward on the Pareto frontier in front of the 2-stage architectures. At the same time, 1-stage architectures (which have the highest reusability ratio of 100%) benefit from reusability even more, forming a separate part of the Pareto frontier starting with the case of 3 uses. Expectedly, the HLS average mass/cost performance increases with the increase in the number of HLS uses for all reusable architectures; these gains, however, tend to decrease with each additional use added.

In terms of the propellant combinations, pure LOX/LH2 or combined LOX/LH2/LOX/CH4 architectures dominate the tradespace; none of the Pareto-dominating architectures use NTO/MMH. Assuming that the inter-stage propellant compatibility is a preferred option for reusable systems refueling from the Gateway, the 1-stage and 3-stage all LOX/LH2 architectures were identified as the likeliest candidates to have lowest HLS-related production and launch costs.

Further cost analysis performed on these two architectures showed that the 1-stage HLS wins over the 3-stage system in terms of the overall HLS program cost if a long-term exploration program (on the order of tens of missions) is assumed. Moreover, the more intensive and longer the program, the greater the cost benefit of the 1-stage HLS.

Thus, the 1-stage LOX/LH2 HLS architecture is the 'ultimate' winner under the assumptions made in this study. It is worth noting, however, that we used a simple HLS model that did not account for the propellant boil-off and, therefore, our analysis did not account for the respective differences in the mass performance of HLS architectures of different propellant combinations – at this stage, we assumed that the mission, lasting only about 10 days, is too short for the differences to have a significant impact on the results of the architectures comparison. Verifying this assumption by exploring sensitivity of the results to the boil-off modeling might be one of further lines of work on this topic.

We also shall note that the results obtained here are based solely on the HLS mass performance and cost aspects; such traditional figures of merit as crew safety, probability of mission success as well as project management risks considerations were left out of the analysis. We assumed that those will require a more elaborate modeling of the system and can be addressed at a later stage.

While we explored sensitivity of the results to the assumption on the exploration program intensity and duration, their sensitivity to the HLS and cost models coefficients was not analyzed. Reusable HLS is, however, a new endeavor being undertaken in the rapidly changing environment of the current aerospace industry. Under these circumstances, it is difficult to predict the accuracy of our models based on the Apollo-era data and cost estimating relationships obtained for traditional spacecraft. This drawback, however, can be rectified by repeating the analysis with the updated model coefficients when and if more accurate data are available.

The results of this study will be of interest to system engineers concerned with architecting and designing future human landing systems. The mathematical models proposed in this paper can be reused and adapted for a range of planetary exploration systems, and are thus of general interest to the space systems engineering community.

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